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SPACE SYSTEMS ENVIRONMENTAL INTERACTION STUDIES

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Under Task 1, the Photovoltaic Array Space Power Plus Diagnostics [PASP Plus] experiment, encompassing most of Amptek's effort in the report period, Amptek has: (1) designed and built the experiment Controller; (2) provided an Electrostatic Analyzer (ESA), two QCMs, and three calorimeters (CALs); (3) worked with PL/GPSP on testing and integrating PASP Plus into the APEX satellite; and (4) provided Ground Support Equipment for spacecraft integration and mission operation. The Controller is comprised of a central processing unit, an array biasing and electrometer unit, an array I-V curve measurement unit, a power conversion unit, and interfaces to the 16 array modules, ten PASP Plus instruments (not including the Dosimeter), and the APEX spacecraft. The PASP Plus Controller and ten other instruments it controls [Sun Sensor, Langmuir Probe, ESA, Transient Pulse Monitor (with sensors), Electron Emitter, 2 QCMs, 3 CALs] are described in detail, giving size (dimensions), weight, data rates and outputs, and input power requirements. On Tasks 2 and 3, at a much lower level of effort, Amptek supported PL/GPSP's work on the Charge Hazards and Wake Studies [CHAWS] experiment (0 to -5000 V sweep power supply and engineering assistance) and on the Space Wave Interactions with Plasmas Experiment [SWIPE] (instrument definition).

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1.0 INTRODUCTION

This is the first annual report for Contract #F19628-91-C-0112, whose objective is to investigate spacecraft-environment interaction and space-plasma dynamics issues, especially as they relate to differential charging, discharge mechanisms, coupling between plasmas and space power systems, wake formation mechanisms, EM wave-particle interactions, and long term radiation effects on operating systems in space.

The work is to be accomplished in three distinct *in-situ* investigatory programs (identified as Task #'s 1, 2 and 3) and the completion of a data analysis and presentation period. Work is currently underway on all three tasks, although the principal effort to date has been expended on Task #1, whose hardware phase is nearing completion. This report presents significant details of that effort and summarizes the progress to date on the other two tasks. The organization is as follows. The objectives of the Task #1 are outlined in the immediately following Section 1.1. A review of the experimental particulars and the science issues involved from Amptek, Inc.'s standpoint are presented in Section 1.2 and Section 1.3 respectively. A brief description of each instrument in the payload is then given in Section 2. Included here is a summary of the unit's function, as well as, its specifications for power requirement, input voltage, input and output signals, data rate, physical dimensions and weight. Specifics on the design of the principal instrument in the payload – the payload Controller – appear in Section 3, while Section 4 contains details on both hardware, software, and the performance of the Ground Support Equipment (GSE), which was developed to support the mission. The real-time GSE display is also addressed in Section 4. The progress to date on Task #2 and Task #3 is covered in Sections 5 and 6.

1.1 PASP Plus Program

The objectives of Task #1 are to be accomplished in concert with the Photovoltaic Array Space Power Plus Diagnostics (PASP Plus) program. This is a Philips Laboratory experiment to be put into orbit on the Advanced Photovoltaic and Electronics Experiment (APEX) satellite. APEX will be placed into an elliptical orbit with a nominal perigee of 350 km (≈195 nm) and nominal apogee of 1850 km (≈1054 nm), by a Pegasus air-launched vehicle, provided by Orbital Sciences Corp. PASP Plus will be used synonymously with Task #1, in all subsequent references.

The objectives of the mission are: (1) to measure the environmental interaction of advanced solar arrays in relation to environmental conditions, when the arrays are subjected to bias voltages up to ±500 V; (2) to determine the long-term radiation degradation effects on various array materials and (3) to flight qualify various array designs after an analysis of the data generated by these two aspects of the investigation.

To accomplish these objectives, the I-V characteristics of 16 array modules from 12 different solar-array types will be measured over the 3-year lifetime of the mission. Additionally, for a period of approximately six months to a year, 10 of the 16 array modules will be biased in voltage sequences between ±500V. Four Electric Field ($\bar{E}$) sensors and a Current Probe of a Transient Pulse Monitor (TPM) will characterize the occurrence of any arc discharge during this time. The payload will also include an ElectroStatic Analyzer (ESA) and a Langmuir Probe to characterize the ambient environment in terms of particle density and energy. Two Quartz Crystal Microbalances (QCM) and three Thermal Coating Calorimeters (CAL) will monitor the surface contamination exposure of the arrays and aid in tracking the magnitude of this factor, which plays a significant role in array performance degradation on orbit. In addition, a Dosimeter (to measure the radiation dosage received at the payload) and an Electron Emitter (to vary the local plasma conditions and widen the local environmental parameters for the mission) will also be flown.

Amptek Inc.'s immediate role in the program is as follows: (1) Design and build the Controller, which is comprised of: a Central Processor unit, an Array Bias and Electrometer unit, an I-V Measurement unit, a Power Conversion unit, and interfaces to ten mission instruments (not including the Dosimeter) and to the spacecraft's Payload Interface Module; (2) Provide the ESA, two QCM's and three CAL's; (3) Integrate all instruments onto the payload, in association with PL/GPSP; (4) Provide a GSE to support spacecraft integration and mission operations; and (5) At a subsequent time, be involved in the analysis of the returned data.

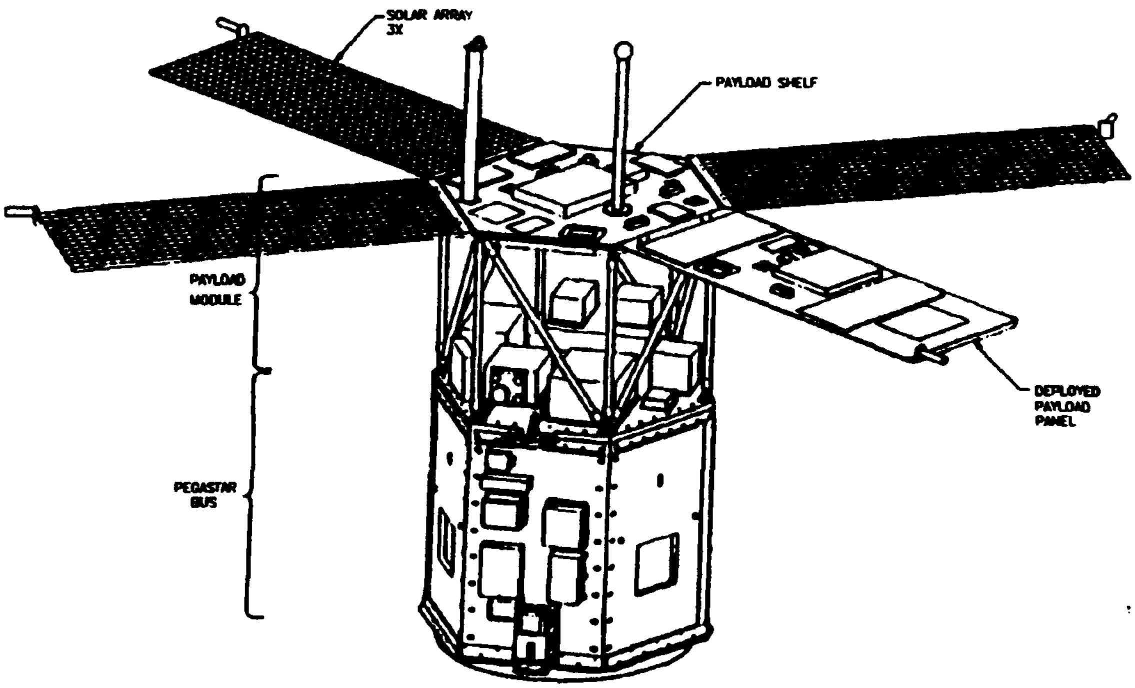


Figure 1: The APEX Satellite

1.2 Experimental Particulars

It is nearly ten years now since a space flight experiment was launched to carry out tasks similar to that of PASP Plus, (that is, to assess the interaction of solar arrays in Low Earth Orbit (LEO) plasmas, to study the nature of the interaction, and to try to improve existing models that attempt to quantify the process). The Plasma Interaction EXperiment II (PIX-II), which was an auxiliary payload on the second stage of the Delta rocket that carried the IRAS satellite into orbit in January 1983, was the last such endeavor. It was preceded by PIX I – from which a scant two hours of telemetry was obtained – some five years earlier. With such a long dearth of *in-situ* data to examine and with the very slim possibility of a similar flight occurring any time soon, PASP Plus (and the welcomed wealth of information that it will provide) could most likely be the defining mission for virtually all issues relating to solar-array plasma interaction issues for quite sometime into the future.

The PASP Plus mission is significantly more comprehensive than PIX-II in its instrument complement. Since it is both sun pointing and spin stabilized, PASP Plus will also be free from the spin-induced probe-shadowing effects which bedeviled the analysis of PIX-II data[1]. Indeed, given the suite of instruments to be taken into orbit, it will be possible to detect and characterize arc discharges, measure ambient particle densities and electron temperatures, array surface contamination, as well as a radiation dosage. PASP Plus is a comprehensive attempt, therefore, to study both the short and long-term environmental interaction effects on solar arrays in LEO.

1.3 Review of Issues Involved

The matter of greatest interest regarding solar array interaction with the ambient space environment and a principal concern of the PASP Plus mission is arcing discharge at high negative operational voltages. The subject received some attention during the 1970's from laboratory studies conducted at NASA Lewis, TRW Inc. and elsewhere[2-4]. Also, as previously mentioned, two very limited *in-situ* investigations were carried out in the PIX missions[5,6]. The issues involved are far from being resolved today, however. The prevailing consensus (if one could be uniquely identified) seems to be that arcing is initiated from some point on an array structure to the ambient plasma when the localized voltage potential exceeds a certain negative threshold – a rather obvious proposition. Details on why the arc starts, such as the threshold level for initiation, the location on an array most favored for arcing, and the relationship between the initiation of arcing and the ambient plasma environment, are all issues of some debate at this time.

In the context of differential charging on spacecraft, Garrett[7] defines arcing to be a rapid (≈ nanosecond) rearrangement of charge by punch through (breakdown from dielectric to substrate), by flashover (propagating sub-surface discharge), blowoff (arc to surface), between surfaces, or between exposed surfaces on the satellite and space. It is certainly possible that most, if not all of these factors, could play some role in arcing on a particular solar array. Given the variety of methods used to fabricate the arrays on

PASP Plus (which range from the concentrator designs, to the bulk Silicon modules, to a thin film GaAs wrap-thru), it might be expected that not only could different factors be at play for the various arrays, but multiple factors could indeed influence arcing on any specific module.

This multiplicity of manifestations has been observed in various investigations of arcing on solar arrays that have been carried out in recent times. Marinelli et al.[8] saw arcing from interconnect leads and exposed adhesives in their study of a 1cm x 2cm silicon cell. Hasting et al.[9], on the other hand, have proposed that the arcing they observed is due to the gas emitted under electron bombardment from the cover glass of the cell. Jongeward et al.[10] suggested earlier that the arcing they saw could have been a consequence of charge buildup on a thin insulating layer over the metallic interconnects, while Ferguson[11], from his work at NASA Lewis, makes it clear that arcing thresholds are a function of solar-cell geometry and materials.

1.3.1 Negative Bias Arcing

From the arc rates observed in chamber tests that were attributed to negative potential discharges, an empirical relationship has been proposed by Ferguson[11]. It is:

$$R = C_1 n \sqrt{\frac{T}{M}} V^x \quad \ldots\ldots\ldots (1)$$

where C_1 and C_2 are constants

n = plasma density

T = plasma temp.

M = ion mass

x is a number based on cell geometry

V = interconnect voltage relative to the plasma potential.

He also proposes that the threshold voltage can be determined as follows:

$$V_{on} = C_2 n^{(-1/x)} \quad \ldots\ldots\ldots\ldots (2)$$

A graph of observed arc rate vs. voltage for standard-interconnect cells is shown in Figure 2. PIX II ground and flight data are also shown for comparison. For whatever their worth, Equations (1) and (2) are the only two relationships in the literature that seek to relate arc rate and threshold for negative charging on solar array interactions in LEO plasmas, .

The PASP Plus data will be looked at to see how well these formulas are corroborated. More than likely, any correspondence will be serendipitous. For, although there is a variable (x) that basically takes in account differences in cell area, there is no accounting for differences in the basic design of array types. It is doubtful, for example, that Equations (1) and (2) will hold for concentrator designs, as opposed to the flat panel-type silicon array modules upon which the formulation data is based.

Array design and fabrication techniques play a significant role in solar array arcing on-orbit; this has also been demonstrated by Martinelli et al.[8] in an investigation into the location and circumstances under which arcing occurs on the previously mentioned Spectrolab array module. It was noted then that upon *properly* cleaning and preparing the

module (essentially improving the quality of workmanship on the unit) the arc rate was reduced by a factor of 100 in frequency. Furthermore, the arc rate showed no dependence on temperature and other factors, as was previously observed.

1.3.2 Positive Bias Concerns

Due to the high mobility of electrons in the infinite reservoir of LEO plasma, arcing is typically not observed under conditions of high positive potentials on solar arrays *in-situ*, nor in laboratory testing generally. An electron leakage current is typically observed to flow between the array and the surrounding plasma. For an ideal collecting surface, this coupling current density (J) is given by:

$$J = n_e e v_{the} \quad \ldots\ldots\ldots\ldots\ldots \quad (3)$$

where $n_e \equiv$ Boltzmann's Relation

$$= n_o \exp(e\phi/KT_e)$$

e = electronic charge

$$= 1.6022 \times 10^{-19}\,C$$

$$v_{the} = (KT_e/m_e)^{1/2}$$

It has been observed, however, that at positive array potentials above a certain threshold (which is material dependant, but is typically on the order of hundreds of volts) a *snapover* effect occurs, in which nominally insulating surfaces on an array (the array cover glass for example) begin to collect electrons from the plasma. Ferguson[11] posits that secondary electrons generated on the insulating surface rapidly migrate over to the conductive bias surfaces to be collected, giving rise to a distinct jump in the collected current in the process. Figures 3 and 4 show the current vs. voltage results obtained for both negative and positive biasing on the two array-specific *in-situ* data sets obtained to date; i.e.,those of PIX I and PIX II.

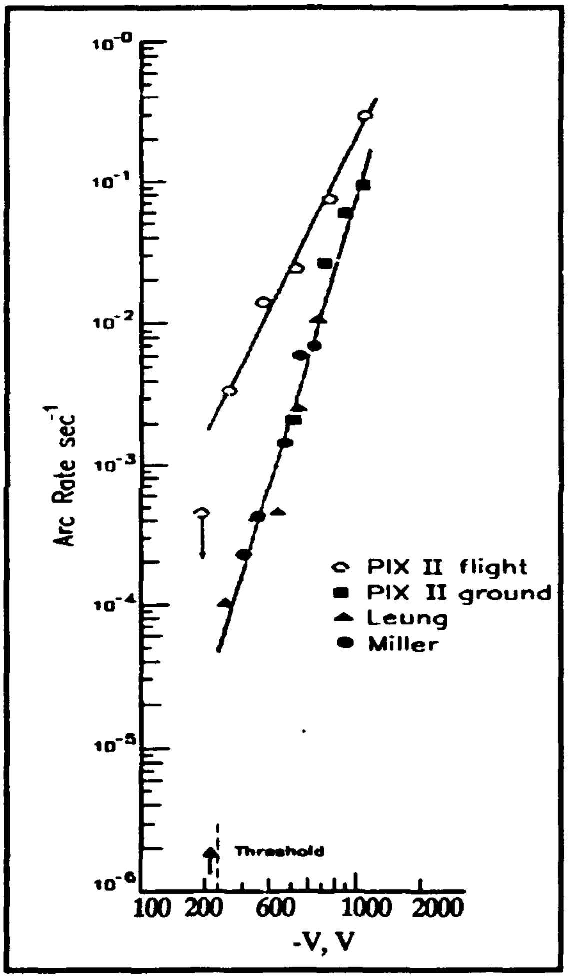


Figure 2: Arc rate vs. Voltage for standard interconnect cells, normalized to LEO ram conditions[11]

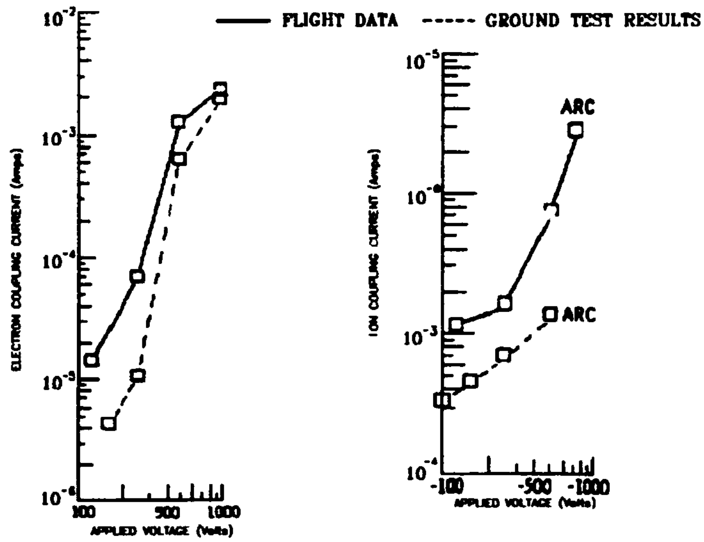


Figure 3: Ground Test and Flight Results - PIX-I

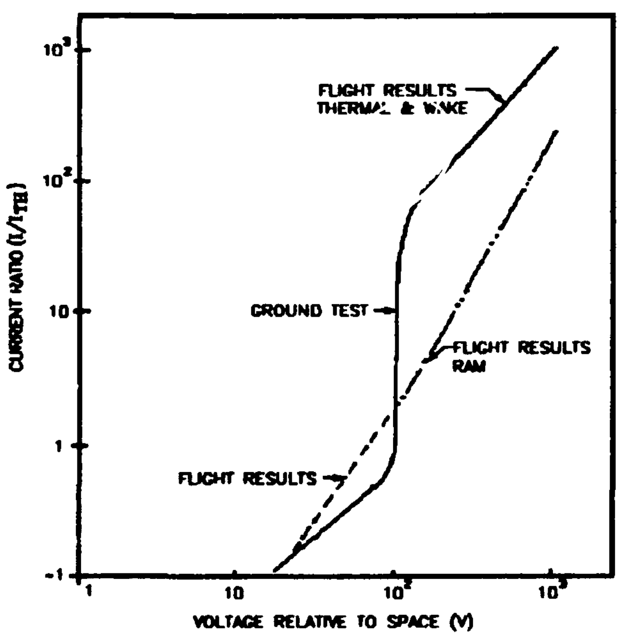


Figure 4a: Ground Test and Flight Results Positive Bias - PIX II

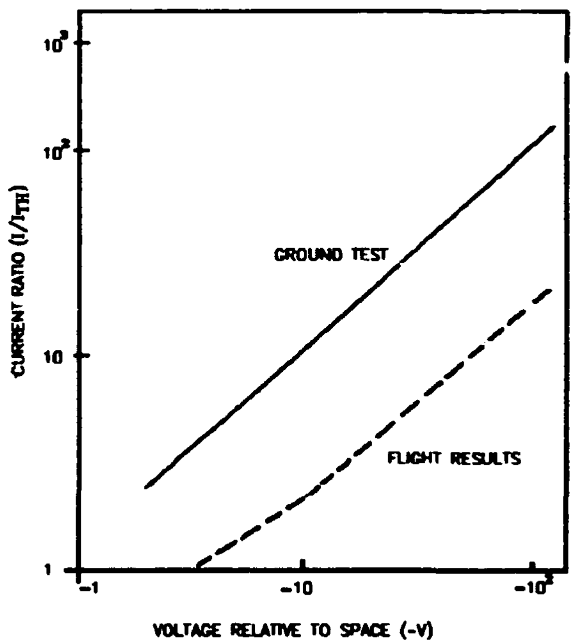


Figure 4b: Ground Test and Flight Results Negative Bias - PIX II

2.0 INSTRUMENT DESCRIPTION

A total of eleven Instruments fall within Amptek Inc.'s purview for this program. They are:

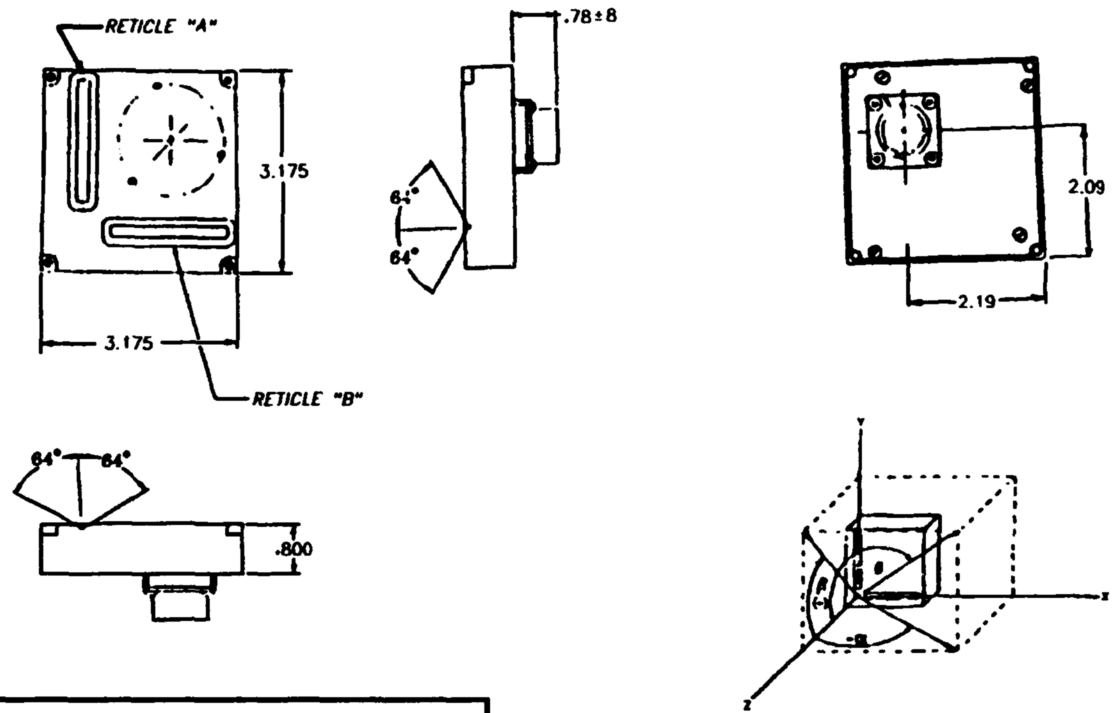
1 - Sun Sensor (SS) and associated electronics.
1 - Langmuir Probe (LP) and associated electronics.
1 - ElectroStatic Analyzer (ESA).
1 - Transient Pulse Monitor (TPM) including sensors.
2 - Quartz Crystal Microbalance (QCM).
3 - Thermal Coating Calorimeter (CAL).
1 - Electron Emitter and associated electronics.
1 - Controller.

We were also tasked with providing all the necessary flight harnesses from the Controller to these instruments, as well as from the solar arrays to the Controller. Except for the Controller, ESA, QCMs and CALs, the other instruments were all supplied by PL/GPSP. A general description of the individual instruments and their interfaces follows.

2.1 Sun Sensor

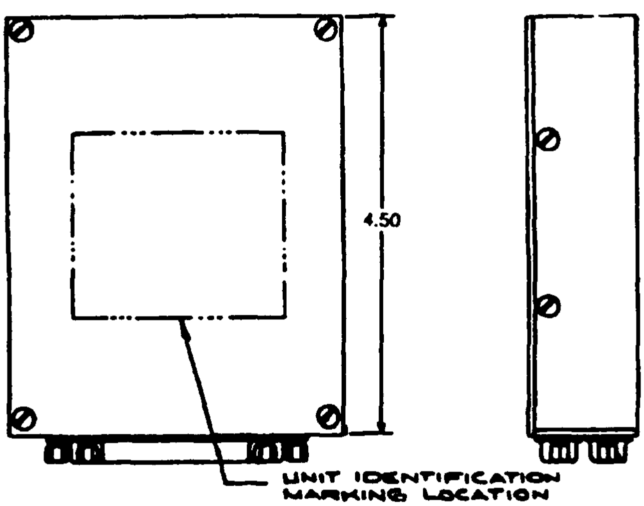
The Sun Sensor will be flown to verify the position of the sun with respect to the Solar Array platforms. The unit is an Adcole Corp. product and consists of a 128° x 128° field-of-view two-axis sensor head and a separate signal processing electronics module. The sun's orientation is sensed by two perpendicular photocell reticules, which generate an analog signal proportional to the intensity of the incident sunlight. The electronics module then outputs a sun angle, in an eight-bit word in a Grey-Coded format. The unit will operate continously during flight. An outline schematic of the Sun Sensor Head is shown in Figure 5a and that of the Electronics Module in Figure 5b. The unit's specifications include:

Power Requirements:	0.7 W Peak; 0.5 W NOM.
Input Voltage:	$+28\,^{+5}_{-4}$ VDC excluding noise and ripple.
Inrush Current:	5 A for 10 μsec; 0.5 A for 10 μsec through 10 msec; 0.1 A for 10 msec through 50 msec.
Outputs:	16-bit parallel Gray Code (8 bits per axis, 0 ↔ +5.12 VDC). one Intensity Threshold Output (1 bit, 0 ↔ +5.12 VDC). +10V supply monitor. (+3.0 VDC NOM.)
Data Rate:	32 bits/sec
Sensor Size:	3.175 in x 3.175 in x 0.8 in
Sensor Weight:	0.63 lb
Electronics Size:	3.5 in x 4.5 in x 1.16 in
Electronics Weight:	0.96 lb



COORDINATE REFERENCE SYSTEM

Figure 5a: Sun Sensor Head Outline Schematic



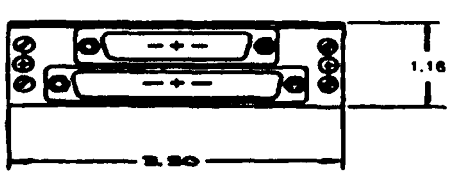
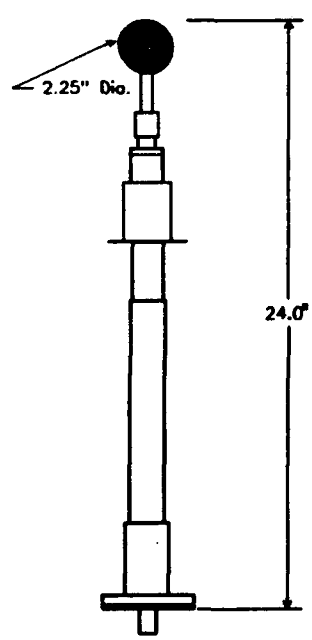


Figure 5b: Sun Sensor Electronics Module Outline Schematic

2.2 Langmuir Probe

The Langmuir Probe was designed and built by PL/GPSG in association with the University of Texas at Dallas, to measure thermal electron density and temperature, as well as the potential of the spacecraft with respect to the ambient plasma. The unit consists of a boom-sensor assembly (an outline of which is shown in Figure 6a) and a signal processing electronics module, whose outline schematic is shown in Figure 6b. The Probe itself is a spherical collector surrounded by a spherical grid. The potential on the collector is swept from +4 V to -4 V with respect to the plasma, and the resulting current vs voltage characteristic is analyzed to obtain the desired information. The unit's specifications include:

Power Requirements:	3.5 W Peak; 1.6 W NOM.
Input Voltage:	28 ± 4 VDC
Inrush Current:	1.2 A for 1msec
Outputs:	three (0 - 5 V) Analog Signals Ne, BIASMON, TEMPMON
Data Rate:	1152 bits/sec
Sensor Head Size:	2.25 in Dia (sphere)
Sensor Head Weight:	0.47 lb
Sensor Boom Size:	2 in Dia. (MIN) x 22 in length; *4 in Dia base*
Sensor Boom Weight:	2.93 lb
Electronics Size:	7 in x 6 in x 4 in
Electronics Weight:	3.97 lb



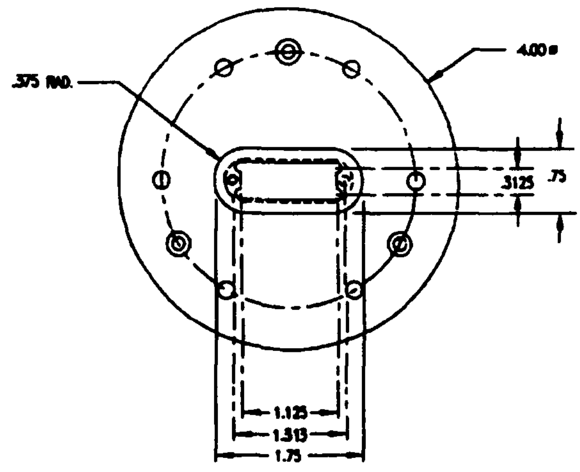
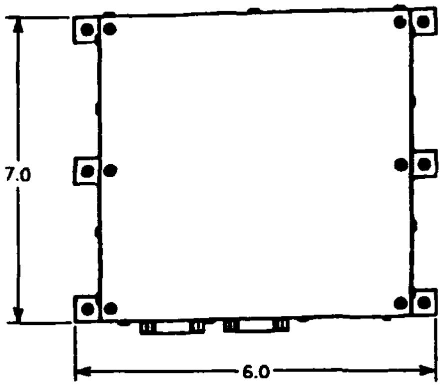
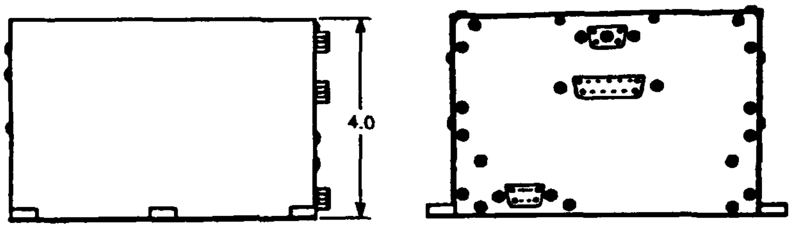


Figure 6a: Langmuir Probe Sensor/Boom Assembly and Boom Footprint

Figure 6b: Langmuir Probe Electronics Module Outline Schematic





2.3 ElectroStatic Analyzer

The ElectroStatic Analyzer to be flown will be provided by Amptek Inc. and is identical to the SSJ/4 detectors supplied to the Defense Meteorological Satellite Program (DMSP). The unit is a single module with 4 incident apertures. It measures the number of ions and electrons within the energy range of 30 eV to 30 keV, in 20 discrete channels. Outline schematics are shown in Figure 7. The units' specifications include:

Power Requirements:	0.5 W NOM.
Input Voltage:	28 ± 6 VDC
Outputs:	4 (5V) Digital Signals;
	EL, EH, PL, PH
	three (0 - 5V) Analog Signals;
	TEMP, HVMON, LVMON
Data Rate:	640 bits/sec for Digital Signals
	240 bits/30 sec data frame for Analogs
Dimensions:	5.75 in x 6.25 in x 6 in
Weight	4.9 lb

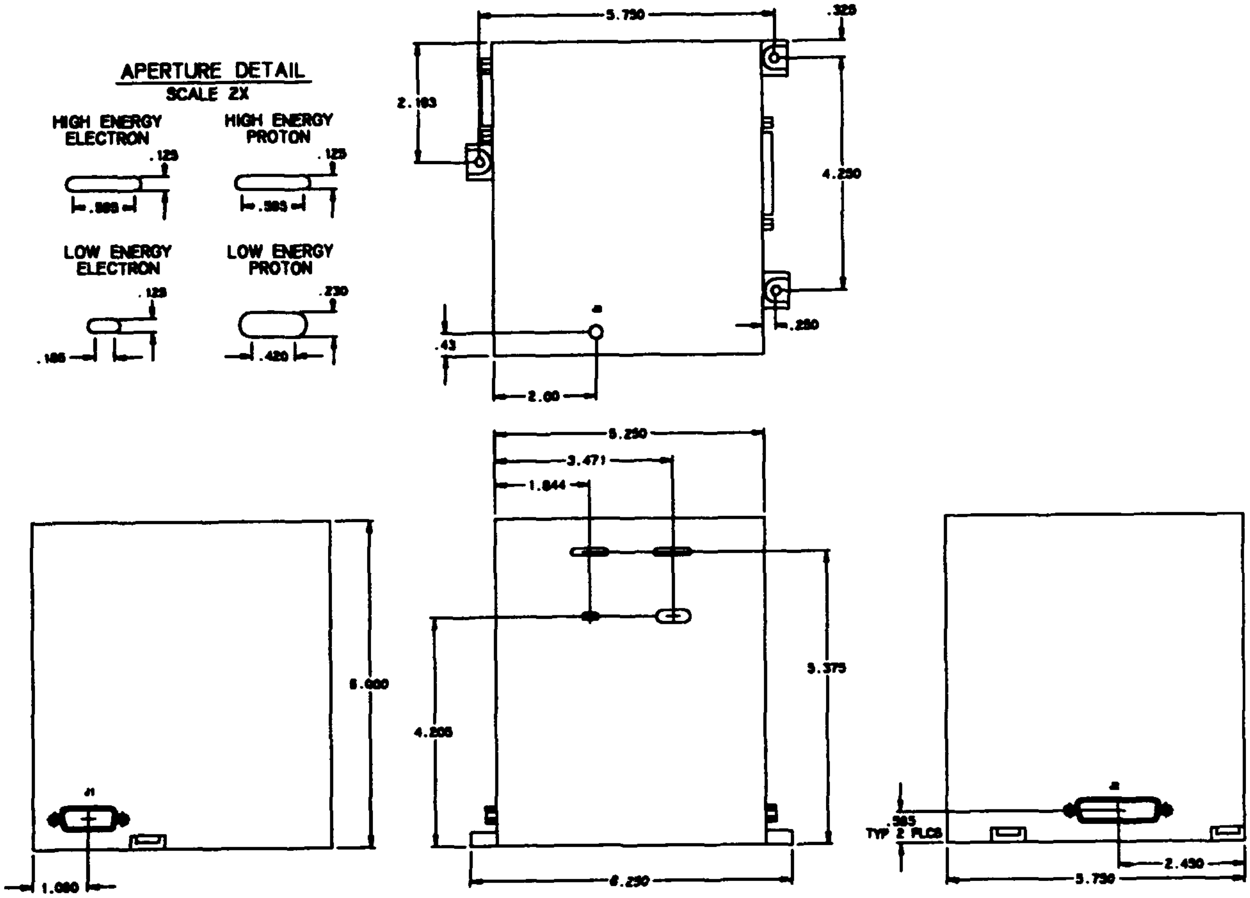


Figure 7: ElectroStatic Analyzer Outline Schematics

2.4 Transient Pulse Monitor

The Transient Pulse Monitor was designed and built by SRI, Intl. to monitor the occurrence and waveform characteristics of arc discharges from the solar arrays in the payload. The instrument determines the pulse rate, peak amplitude, rise time and pulse integral of transient electromagnetic signals, by processing the input of up to six Electric Field ($\bar{E}$) and Current Probe sensors. Four $\bar{E}$ Sensors and one Current Probe will be flown. Outline schematics of the instrument's electronics module is shown in Figure 8a. Outline schematics of the $\bar{E}$ Sensor and Current Probe are shown in Figures 8b and 8c respectively. The TPM's specifications include:

Power Requirements:	13.5 W MAX; 12.1 W NOM.
Input Voltage:	28 ± 12 VDC
Outputs:	11 (5V) Digital Signals
	DATA 0 - 7, MR, SI, DR
Data Rate:	352 bits/sec
Dimensions:	
Electronics Unit	14.34 in x 8.83 in x 4.41 in
$\bar{E}$ Sensors	3.49 in x 1.30 in x 0.83 in (each)
Current Probe	2.52 in x 0.51 in x 0.51in
Weight:	
Electronics Unit	13.03 lb
$\bar{E}$ Sensors	0.19 lb (each)
Current Probe	0.19 lb

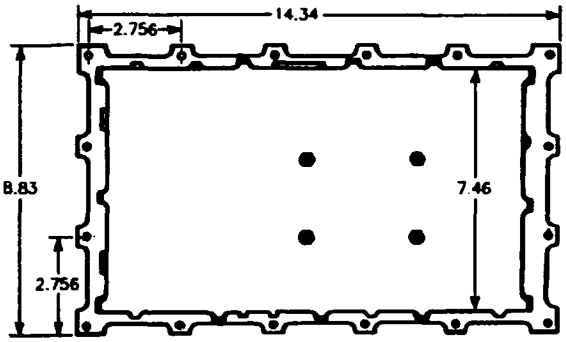
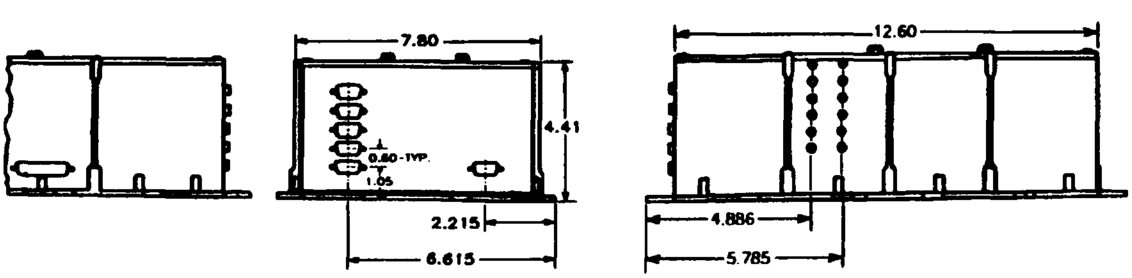


Figure 8a: Transient Pulse Monitor Electronics Module Outline Schematic



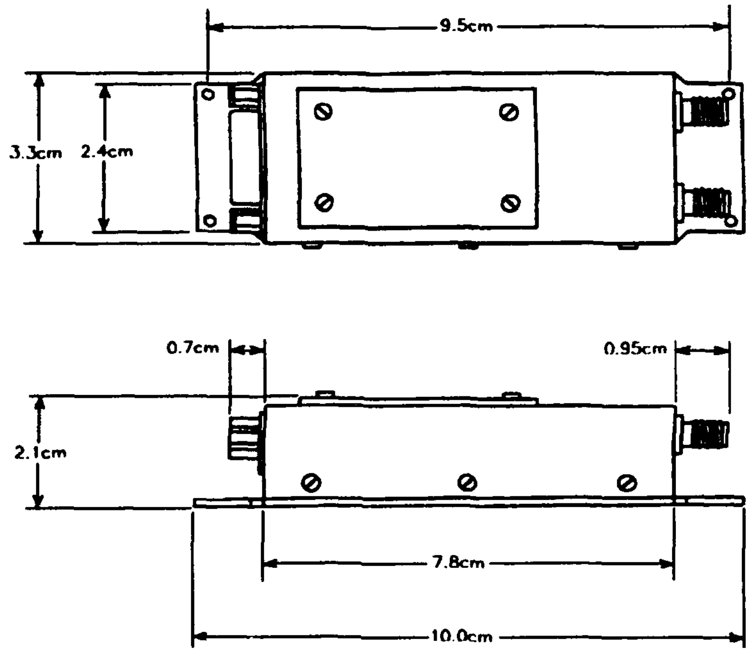


Figure 8b: Transient Pulse Monitor
$\bar{E}$ Sensors Outline Schematic

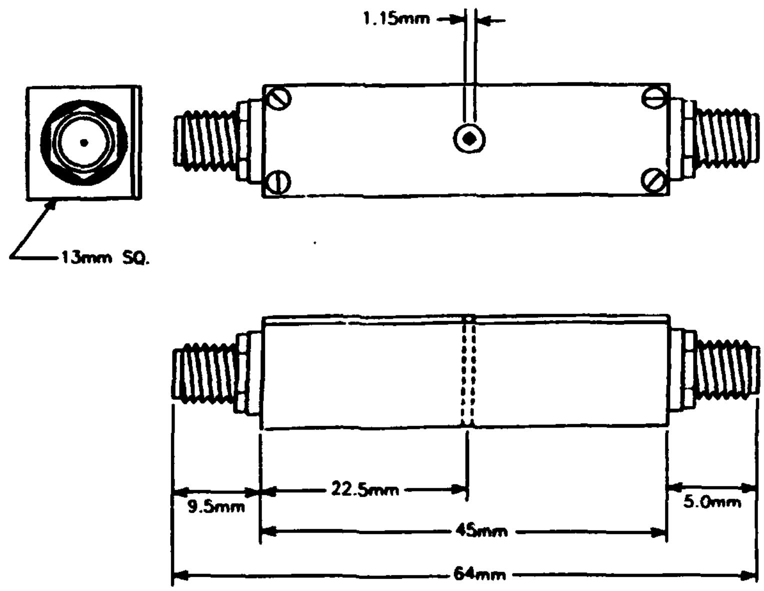


Figure 8c: Transient Pulse Monitor
Current Probe Outline Schematic

2.5 Quartz Crystal Microbalance

The Quartz Crystal Microbalances (two of which will be flown) were designed and built by Faraday Labs to measure the accretion of surface contaminants on orbit. The units consist of a sensor head and a controller. The sensor head is made up of a matched pair of specially cut quartz crystals, each resonating at approximately 15 MHz. One crystal is the designated sensor, the other provides a reference signal. This signal is displaced about 1 kHz above the sensor frequency. When both signals are mixed a beat frequency results that is dependant solely on contaminant accretion. This beat frequency output is extremely sensitive: a 1 Hz shift corresponds to 1.56×10^{-9} g/cm^2 of contaminant deposition. The cross-sectional schematic of a device is shown in Figure 9. The QCM's specifications include:

Power Requirements:	
Oscillators:	30 mW NOM.
Heater:	1.25 W NOM.
Input Voltage:	
Oscillators:	15 ± 0.15 VDC
Heater:	28 ± 4 VDC
Outputs:	one (14 V, 1 - 20 kHz) Digital Signal one (0 - 5 V) TEMP Analog Signal
Data Rate:	56 bits/30 sec data frame
Dimensions:	1.75 in Dia. x 2.42 in (each)
Weight	0.53 lb

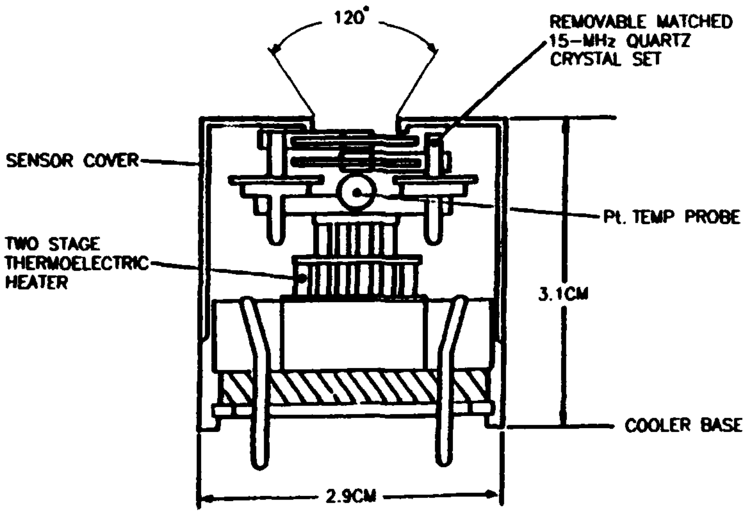


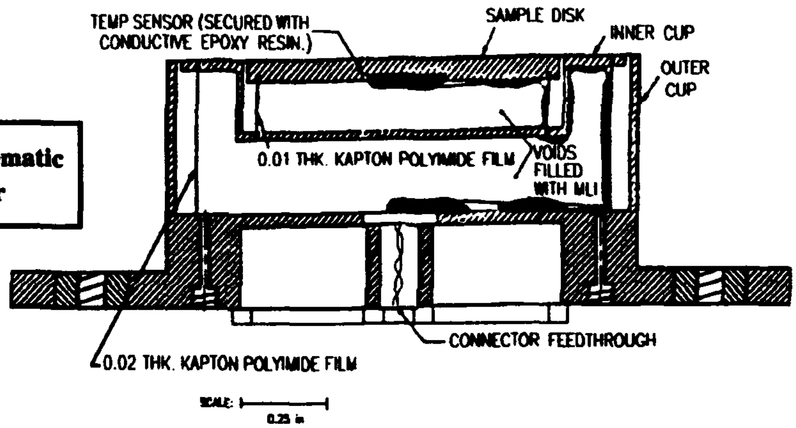
Figure 9: Cross Section Schematic of QCM Sensor Head

2.6 Thermal Coating Calorimeter

The Thermal Coating Calorimeters (three of which will be flown) are supplied by SAIC, Inc. The units consist of a sensor head and a thermometer. The surface coating of each head is identical to the cover glass of a particular flight array. A change in the surface state of a unit will be reflected in a temperature change and is a direct indication of a change in the unit's ratio of absorptivity (α) to emissivity (ε). Along with the QCMs, the CALs will serve as contamination monitors for the mission. A cross-sectional schematic of a device is shown in Figure 10. The units' specifications include:

Outputs:	one (0 - 5V) TEMP Analog Signal
Data Rate:	36 bits/30 sec data frame
Dimensions:	1.87 in x 1.87 in x 0.61 in
Weight	0.41 lb

Figure 10: Cross Section Schematic of a Calorimeter



2.7 Electron Emitter

The Electron Emitter was designed and built by PL/GPSP. It is essentially an electron filament source which is to be operated for a limited period on orbit, in order to widen the local environmental plasma parameters, for the studies to be carried out on orbit. An outline schematic of the unit is shown in Figure 11. The unit's specifications include:

Power Requirements:	35 W NOM.
Input Voltage:	28 ± 6 VDC
Outputs:	Seven (0 - 5V) Analog Signals Emission Mon., Bias Mon., Htr I Mon., Grid I Mon., A/B Mon, LV Mon, Temp Mon.
Data Rate:	12 bits per 30 sec Telemetry data frame for Temp Mon., 48 bits/sec total, for others
Dimensions:	8 in x 6 in x 4 in
Weight	7.14 lb (tot)

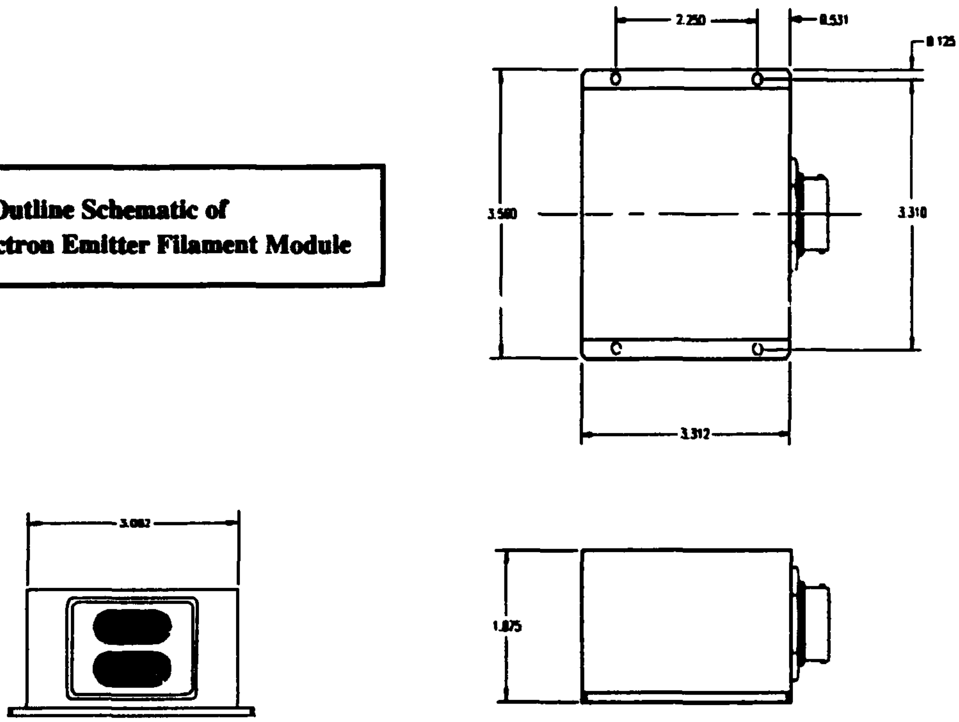


Figure 11a: Outline Schematic of Electron Emitter Filament Module

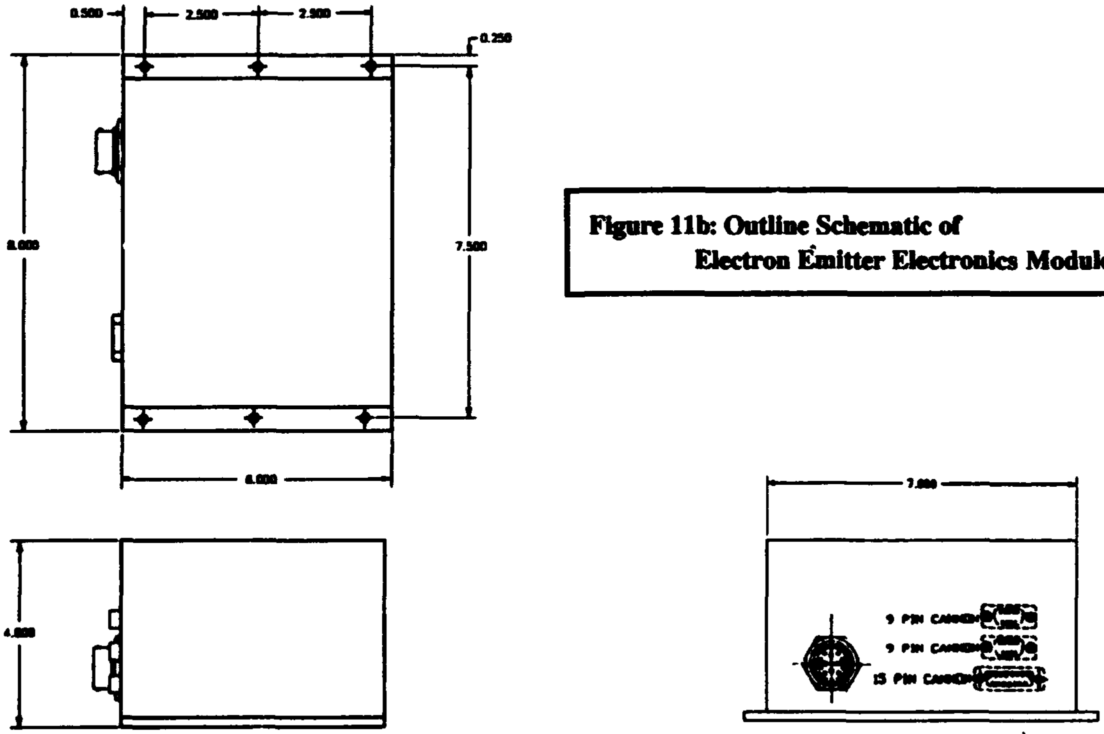


Figure 11b: Outline Schematic of Electron Emitter Electronics Module

2.8 PASP Plus Controller

The Controller was designed and built by AMPTEK Inc. It is the true heart of the PASP Plus payload and is comprised of the following subsystems: (a) a Central Processing unit; (b) an Array Bias and Electrometer unit; (c) an I-V measurement unit; (d) a Power Conversion unit; (e) interfaces to sixteen solar array modules, to ten instruments in the payload complement, and to the spacecraft's Payload Interface Module. The Controller monitors the performance of the arrays by measuring their I-V curves continually. An individual array module can also be biased to any value between ±500V. This might be accomplished by using a predefined bias sequence or by uploading specific values from the ground. In addition, the Controller will control and/or acquire data from all the instruments in the payload. An overall schematic of the instrument is shown in Figure 12. The Controller interfaces are as follows:

Power Requirements:	7.6 W NOM.
(without other units powered on)	
Input Voltage:	28 ± 4 VDC
Signal Link (to s/c):	RxD-; TxD-; RxD+; TxD+.
Data Rate:	19.2 kbits/sec
Dimensions:	9.5 in x 10.0 in x 6.87 in
Weight	12.8 lb

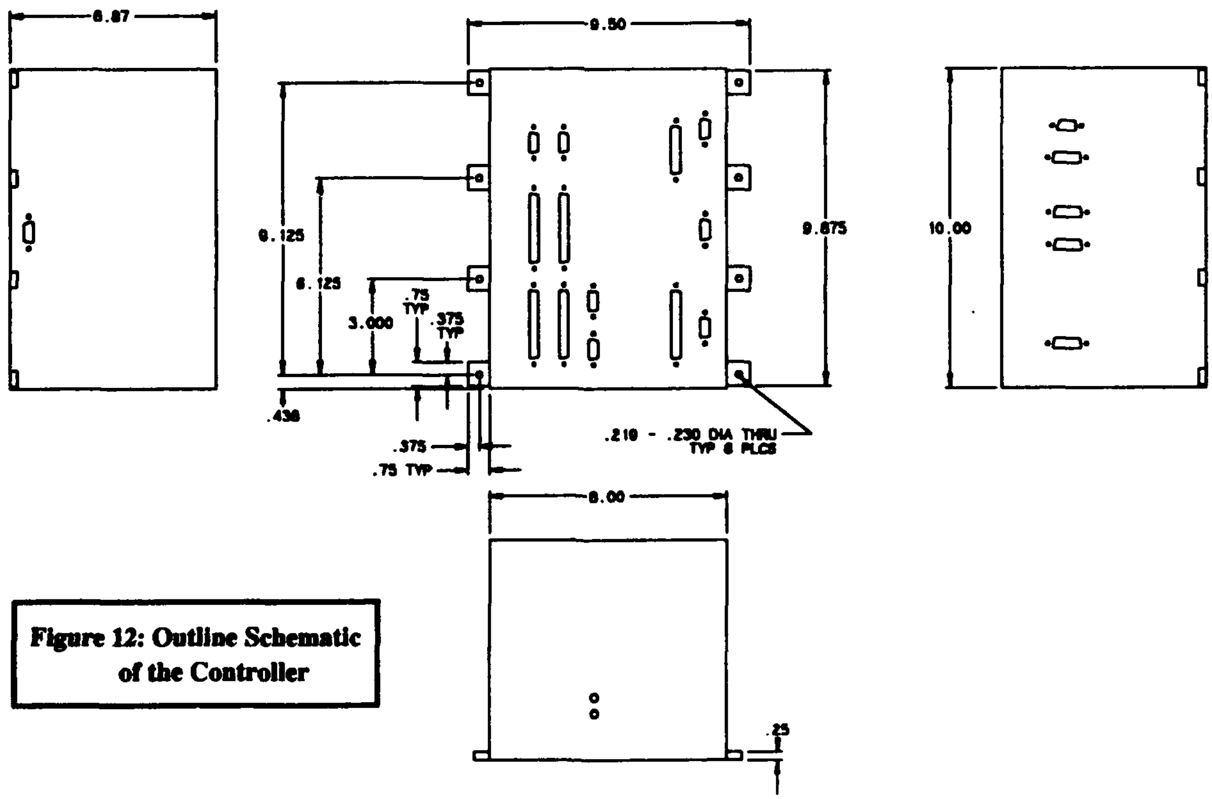


Figure 12: Outline Schematic of the Controller

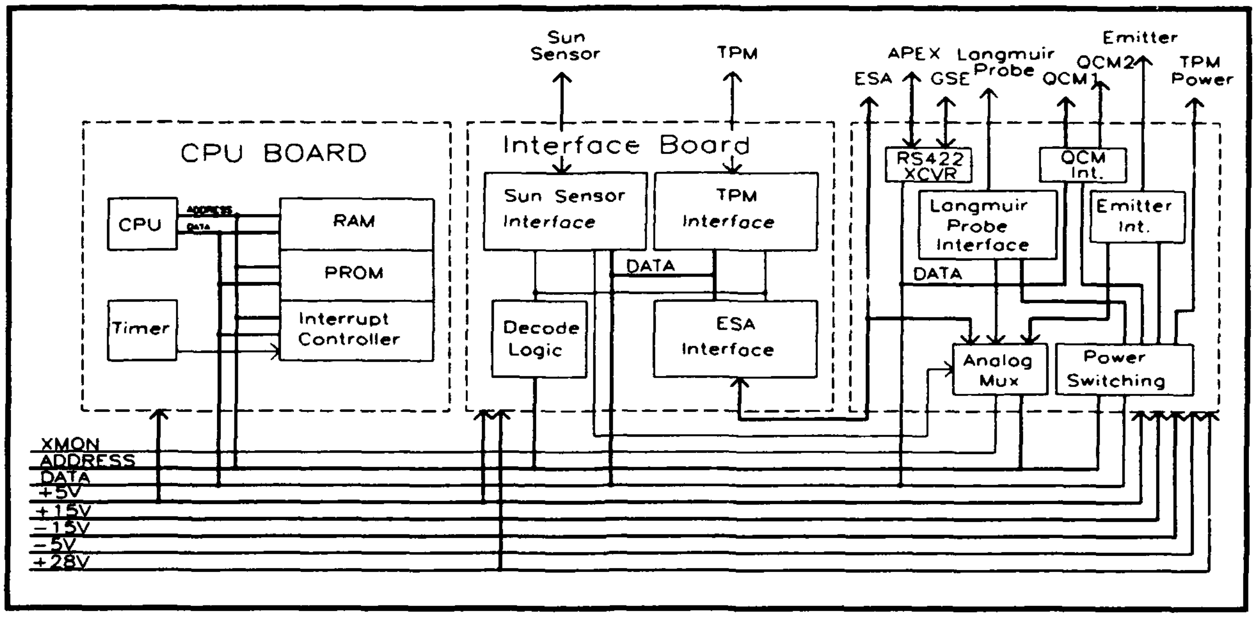


Figure 13: Overview Schematic of Controller's Instrument Interfaces and Digital Processing Units

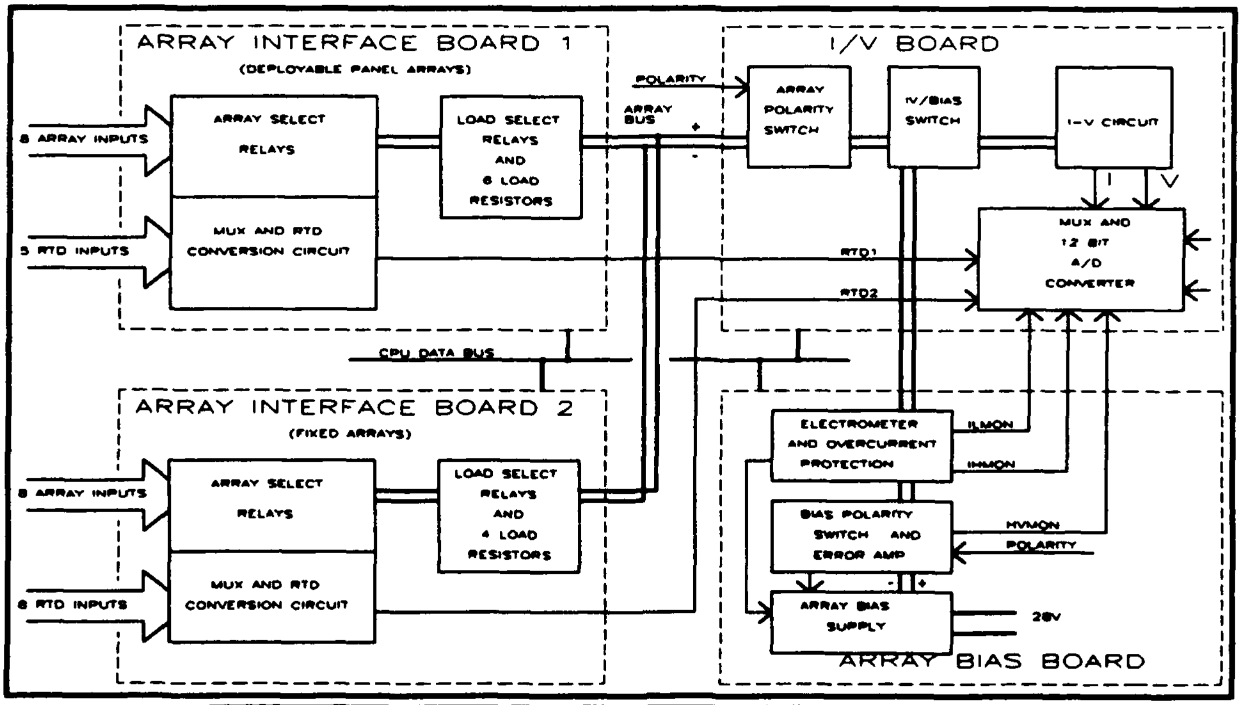


Figure 14: Overview Schematic of Controller's Array Interface Units

3.0 CONTROLLER DESIGN

Overview schematics of the Controller's instrument interfaces and Digital Processing units, as well as the Array Interface units, are shown in Figures 13 and 14, respectively. Both illustrate how the principal functions of the Controller are facilitated in hardware. Further insight into this aspect of the Controller's design is given in Section 3.1. The software aspect of the design is next touched on briefly in Section 3.2.

3.1 Hardware

The I-V measurement subsystem of the Controller is central to the array characterization tasks of the mission. The design required the following to be facilitated: (a) obtain a 64 point sample of current (I) and voltage (V) for each of the 16 array modules; (b) apply a bias potential of between ±25 V and ±500 V to any one of 10 modules; (c) measure array plasma leakage current from 1 μA to 20 mA, with adequate overcurrent protection in the positive bias regime, and (d) measure the temperature of 12 array modules. The design approach taken to satisfy the I-V measurement requirement was to measure an array module's short-circuit current and open-circuit voltage first. A programmable active MOSFET load was then used to set the load resistance at 64 predefined values, and the module's current and voltage was measured for each resistance. Both of these values were measured with 12-bit resolution and the measurement circuitry was optimized to allow minimum series resistance in the array measurements. A schematic of the essential hardware is shown in Figure 15.

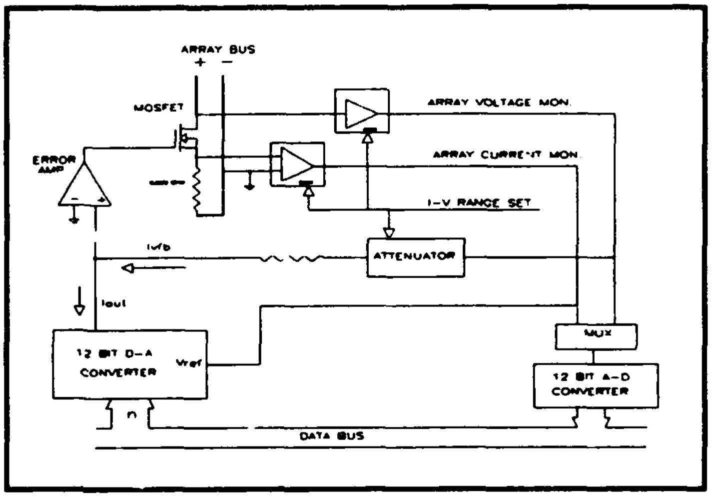


Figure 15: I-V Circuitry Schematic

To satisfy the biasing requirements of the mission, a separate relay-switched load resistor for each of the 10 biased modules was utilized. This allowed the possibility of biasing each of the biased

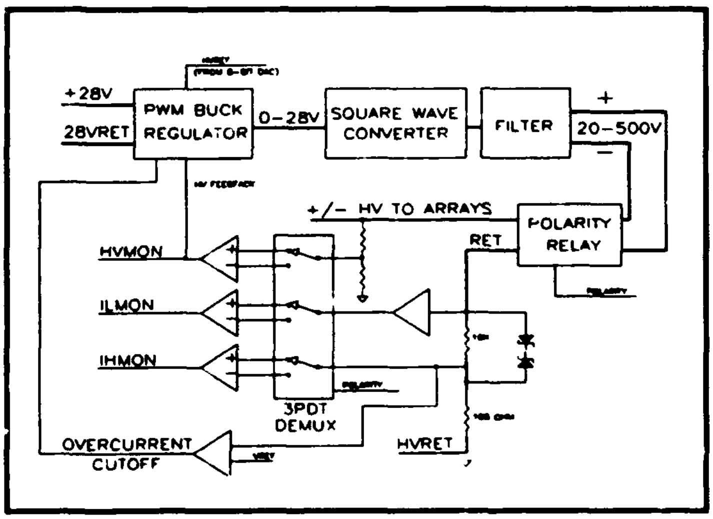


Figure 16: High Voltage Bias Circuitry Schematic

was utilized. This allowed the possibility of biasing each of the biased modules in either an open-circuit, short-circuit or optimum-load configuration. The latter mode is of particular significance, since represents the loaded state of an array, as it would be on-orbit when environmental interaction phenomena occur. Array biasing was actually accomplished by using a MOSFET buck regulator followed by a single MOSFET converter. Leakage current is measured in two linear ranges, to allow good resolution across the variation expected from the different modules. A schematic of this portion of the hardware is shown in Figure 16.

A major objective of the PASP Plus mission is to investigate the impact of radiation on array performance. It was vitally necessary, therefore, that the Controller itself be not unduly susceptible to radiation effects. All electronic parts used in the unit were Rad-Hard to greater than 10^5 Rads total dose, which, given the anticipated environmental parameters, should be more than adequate for the mission duration. In instances where the radiation integrity of a part was not known *a priori*, it was tested at a Radiation Testing facility to ensure that it was suitable for this particular task. The microprocessor choosen for the unit is the Sandia SA3000. It is a flight proven, Rad-Hard and high Single Event Upset (SEU) threshold device, with considerable flight heritage. It is equivalent to Intel's 8088 commercial processor. While it no longer represents state-of-the-art technology, its radiation integrity is unsurpassed. It is also more than adequate for the processing demands of the mission.

3.2 Software

The software which runs on the SA3000 and facilitates the various tasks that the Controller carries out is written in Assembly Language and has within it four identifiable

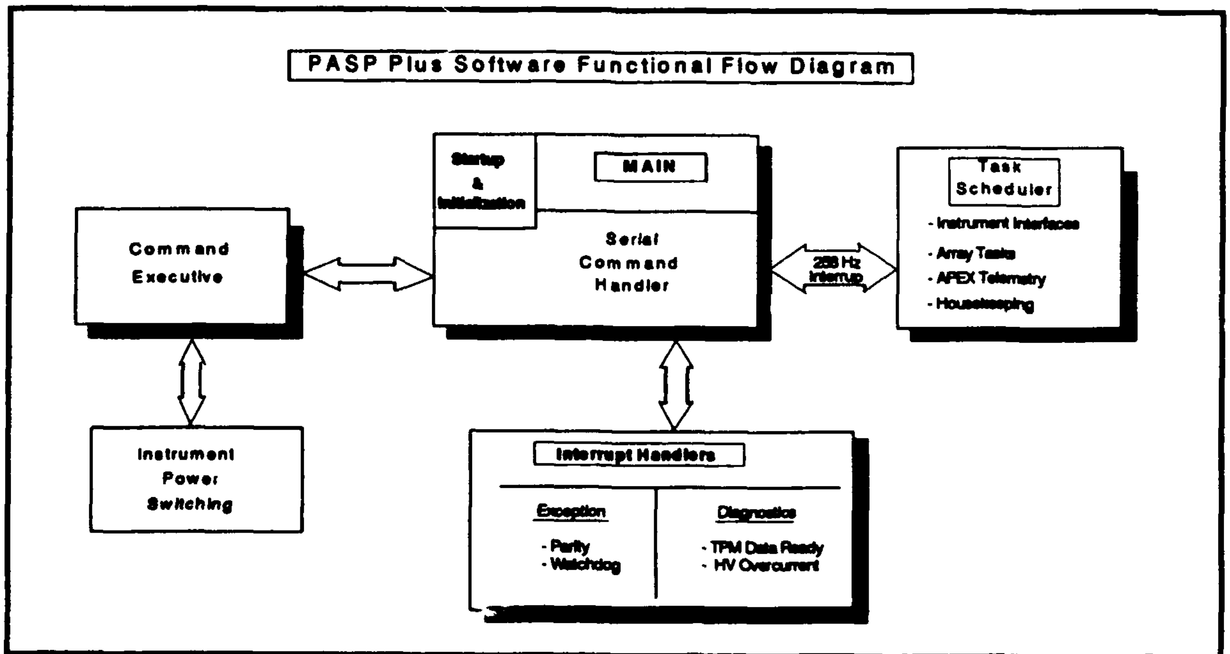


Figure 17: Controller Software Functional Flow Diagram

sections. These are: (a) a main section which contains the Startup and Initialization routines, as well as the Serial Command Handlers necessary for communicating with APEX; (b) a 256 Hz interrupt-driven Task Scheduler (which orders all ongoing activity); (c) an Interrupt Handler routine whose principal task is to service Exception Interrupts (such as Parity Error or Watchdog timing flags) and Payload Diagnostic Interrupts, and (d) a Command Executive section that is concerned with executing payload instrument commands received by the Controller. Figure 17 indicates the functional flow of the PASP Plus software.

4.0 GROUND SUPPORT EQUIPMENT

The PASP Plus Ground Support Equipment (GSE) consists of both hardware and software that were developed specifically to accomplish two tasks. These are: (a) to strip out the PASP Plus data and other spacecraft housekeeping information from the APEX telemetry stream and make it available for display on two instrument-specific display screens, and (b) to display real-time instrument-performance information when the spacecraft is in direct contact with the ground. Additionally, the GSE will carry out rudimentary analysis of the historical instrument performance data, which was stored on-orbit in the interval between ground contacts. Some detail on both of these tasks follows.

4.1 Hardware

The hardware that strips out the required information from the spacecraft downlink and makes it available for the instrument-specific screens is referred to as the PASP Plus Spiderbox. It is actually a printed circuit board which plugs into a slot in a 486/33 IBM type PC. Figure 18 below shows the overall architecture and varous components of the unit.

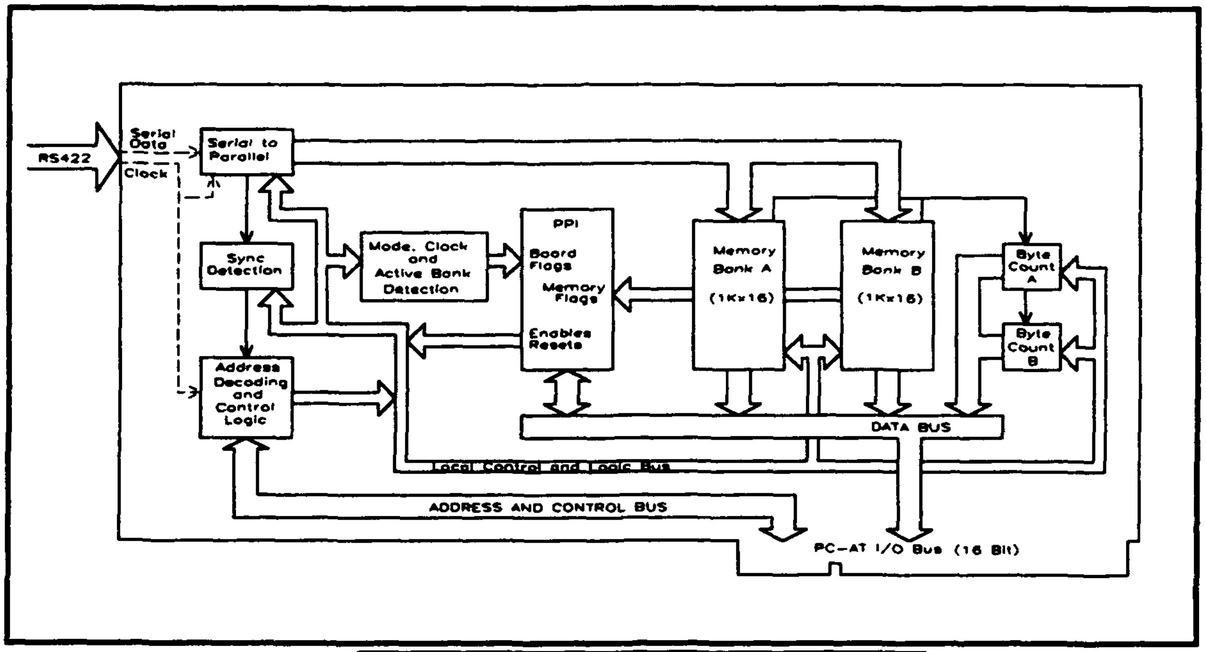


Figure 18: Spiderbox Overview Schematic

In operation, the nominal 128 kbaud RS-422 telemetry feed is examined for the identifying 4-byte synch word placed at the begining of each telemetry minor frame. Once this has been accomplished, data in several of the sixteen minor frames that make up a major frame is retrieved. Some of this information is then made available to the PASP Plus GSE display that is concerned with radiation dosage (from an instrument supplied by a another vendor and, consequently, not a matter for discussion here), while the rest is made

available to the principal GSE display, which displays information for all the other instruments and systems. All of the telemetered data is stored on the local hard disk.
The Spiderbox hardware can also accommodate the APEX contingency telemetry mode of 16 kbaud, as well as signal inversion on both data and clock input lines.

4.2 Software

GSE software was written to run on the PC hosting the Spiderbox to allow it to perform the tasks outlined above. Yet to be done at this time, is the writing of other software, which would allow some analysis of the telemetered data to be carried out post-pass. A full determination of what is to be done in this regard, as well as how it is to be carried out, is still outstanding.

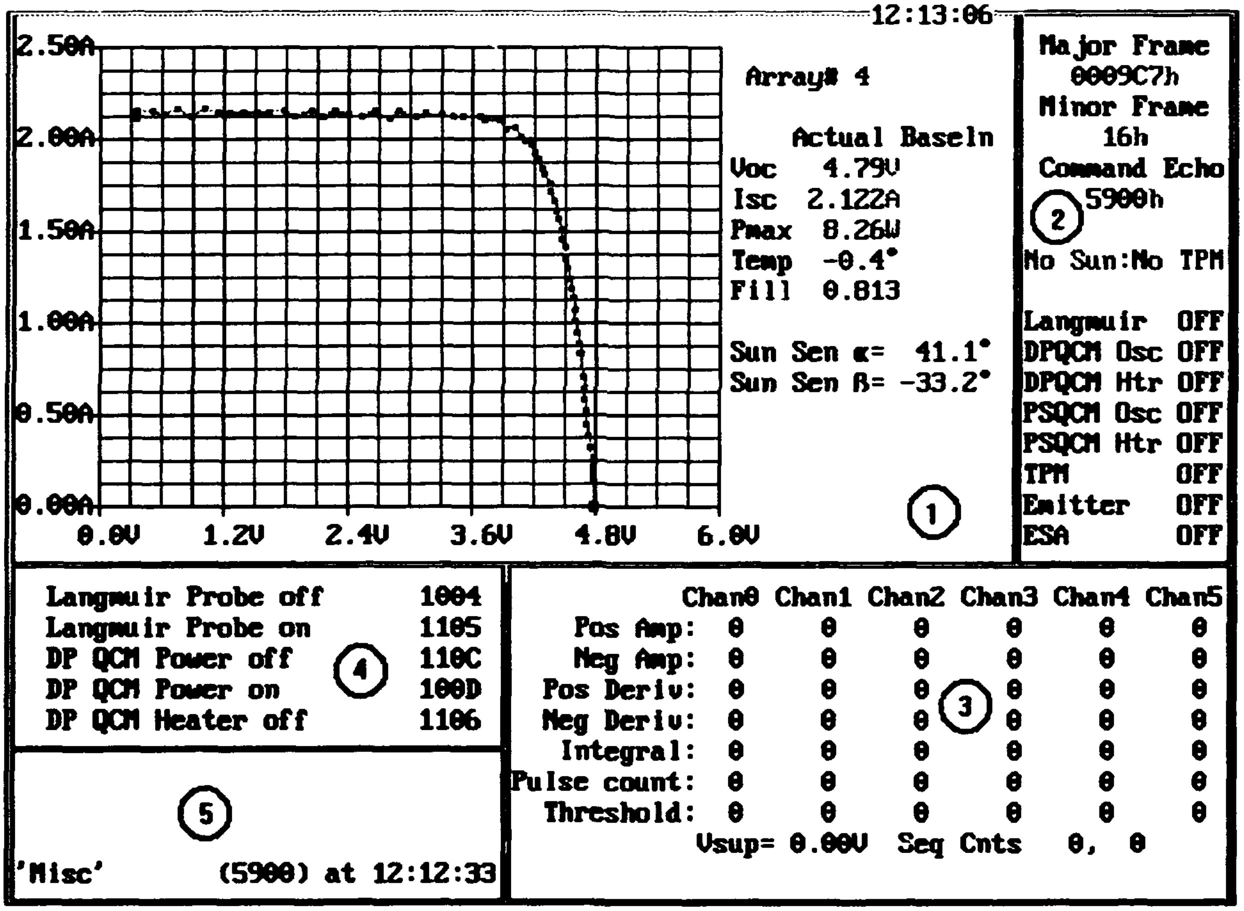


Figure 19: Main GSE Display

Virtually completed, however, is the principal GSE software, which allows realtime data to be displayed during ground contact periods, for all of the PASP Plus instruments and subsystems, with the exception (as was previously mentioned) of the Dosimeter. The code which carries out all of the necessary tasks is the matured version of the developmental software used to exercise and test the Controller. Tailored versions of the

software was used to accomplish specific program test along the way, such as the Thermal Vacuum, EMI/EMC and Thermal Balance/Space Simulation tests.

The main GSE display is shown in Figure 19. It is the default screen on instrument turn ON and, as can be seen, is segmented into five sections. Section ① is the primary array parameter display block. As is illustrated, the I-V characteristic of an array can be viewed and key parameters (Voc, Isc, Pmax, TEMP and Fill) are indicated. On depressing a function key on the keyboard, array temperatures can be seen over a three-hour period, in color-coded graphical form. Also available in a keystroke are all of the housekeeping parameters of the Controller. These too are color coded, so that instrument performance within the green, yellow, or red limits are immediately discernible. The Sun Sensor angles α and β, are also shown in this block. Whether or not the sun is in view is indicated in Section ②. The indicator for whether or not the TPM is ON is also here, as is the status indicator of all the other commanded instruments in the payload. At the top of this block, both the major and minor counters of the incoming telemetry frames are indicated. Section ③ is an instrument-specific display block. The instrument parameters for the TPM are currently displayed here, but with a keystroke the instrument parameters for the Emitter, the Langmuir Probe, or the ESA can be shown. Section ④ is the Command section of the display. Each instrument command can be executed from here. This GSE feature will not exist in the final version of the software that supports mission operations, as *instrument commanding* on orbit will be not be initiated from this display. Finally, Section ⑤ is a command echo block. All command received at the Controller are displayed here, in order received. For mission support, the display area for these will include that of Section ④, with the commands scrolling upwards into this block.

4.3 Performance Summary

In an effort to establish baseline data on the functionality of the array modules in the payload, all sixteen modules in their flight configuration were connected to the Controller and placed in a solar vacuum simulator and temperature cycled in a manner similar to that experienced on orbit for several orbits. No array biasing was initiated in this instance, and the diagnostic instruments were not used, but the Controller's I-V measurement circuitry was thoroughly exercised. In general, the hardware performed superbly over the two week duration of this test. Some minor hardware shortcomings did surface, but these were easily rectified.

The most illuminating outcome of the exercise was the performance of the array modules themselves, over the temperature excursions that were experienced. The variety in characteristics is shown in Figure 20, for fifteen of the modules. The average temperature of the modules is 45°C. It is seen that the variation in short-circuit current ranged from 2.5 A to less than 80 mA, while the open-circuit voltage varied between 33 V and 2.1 V.

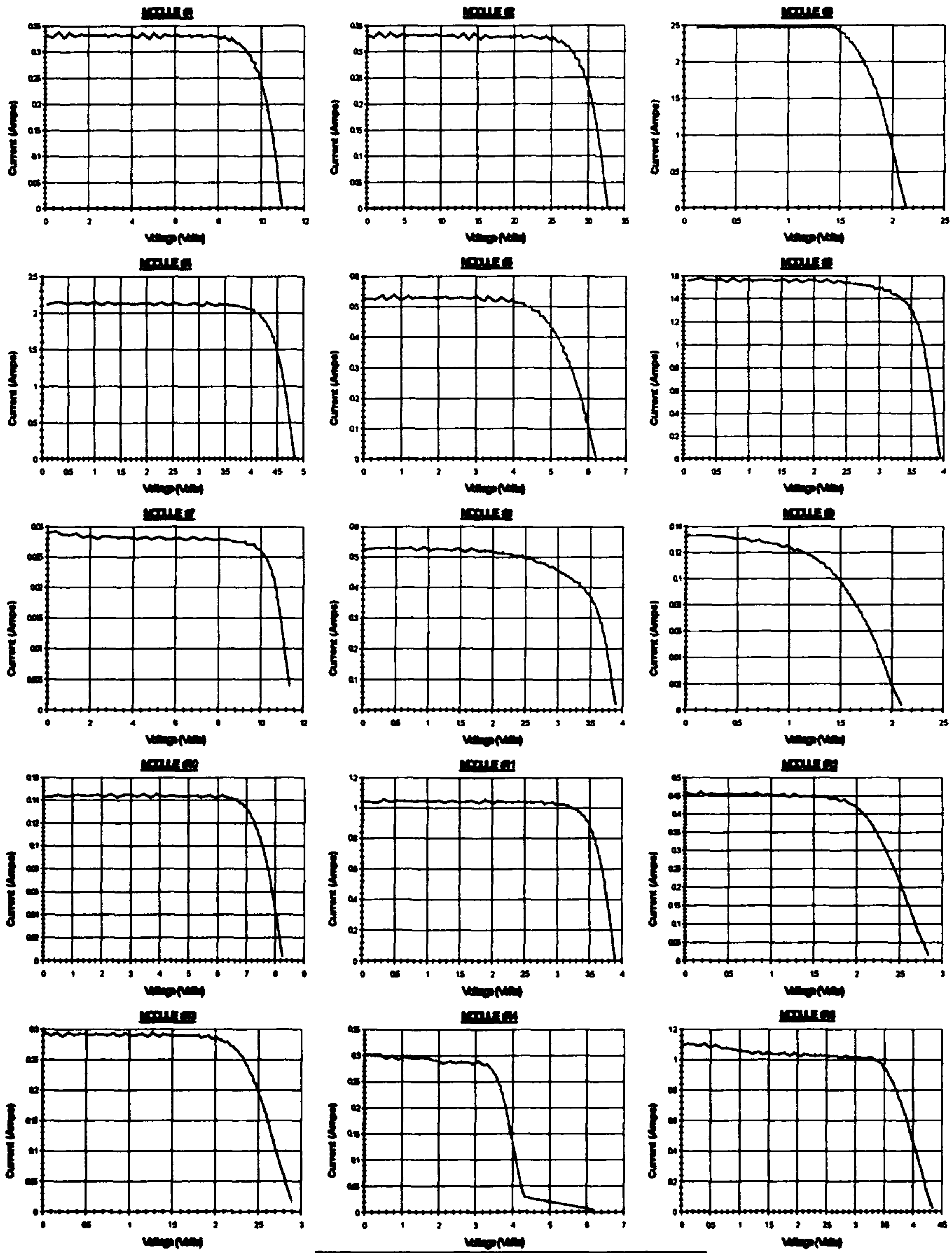


Figure 20: Typical Array I-V Curves

5.0 TASK #2

Task #2 is a study of wake related issues in the ionosphere. An instrumentation package has been assembled to fly as part of the Wake Shield Facility (WSF) Program on the Space Shuttle. The package is designated as the Charge Hazards and Wake Studies (CHAWS) experiment. It consists of several unique Retarding Potential Analyzers (RPAs) mounted in two separate housings. One housing is cylindrical and can be biased with a controlled high voltage which may be swept to potentials as high as minus 5 kilovolts. It is anticipated that such an intense electric field will attract ions from the ambient plasma even when the potential is shielded in the wake created by the WSF structure. A part of the experiment is designed to help characterize this process of collapsing a plasma into the void generated by a structure. Understanding the dynamics of such a process will assist in designing space flight hardware, such as large solar arrays, that may be shielded from plasma by utilizing wake effects.

The principal experiment on WSF is a Molecular Beam Epitaxy (MBE) system that uses the ultra-high vacuum (UHV = $< 1 \times 10^{-12}$ Torr) conditions generated by the WSF to deposit ultra-high-purity silicon on sample wafers and controlled doping of some of these wafers. This is intended to demonstrate the feasibility of spaceborne manufacturing facilities for high-grade semiconductors. The CHAWS experiment contains several RPAs that may be swept from 0 to 32 volts. Such an ion RPA facing into the ram direction can use the orbital velocity signature to resolve mass information from ion energy scans. The orbital velocity of ≈ 7.5 km/sec far exceeds the thermal velocities of the ambient plasma population. Thus all ions appear to have the same velocity with respect to the orbiting platform, and their energy then becomes a function of their mass. The ram-side mounted CHAWS sensors will be able to provide some mass characteristics of the ambient plasma. The wake-side RPAs will monitor the deposition process of the MBE experiment and provide some data as to the amount of perturbation of the environment that the process causes.

The need for maintaining an UHV environment requires all of the wake side hardware to be UHV compatible including the CHAWS experiment. Some of Amptek's staff has had previous experience with UHV design. Amptek has provided technical guidance and support for the design of the wake-side portion of CHAWS. The resulting sensor is constructed entirely of stainless steel, glass, and ceramic. It is bakable to 200°C and is mounted on a metal-sealed UHV flange that attaches to the WSF.

The RPAs that mount inside this probe contain microchannel plates (MCPs) to amplify the collected ions. Amptek has provided expert technical support and guidance in the design, construction, and testing of these unique RPAs. We helped solve the manufacturing problems of construction using only stainless steel, glass, and ceramics. We

assisted in signal routing to isolate the high-voltage biasing system from the low-level signals derived from the MCPs. We suggested a method of using a matrix of countersunk drilled holes to function as a light trap to aide in the solar rejection of the RPAs. We provided guidance as to the selection and handling of MCPs to obtain maximum performance. We provided assistance in the calibration and evaluation of the completed detectors. Amptek is also the manufacturer of the charge sensitive preamplifiers (A111F) used to transform the RPA outputs to digital signals.

Amptek provided a digitally controlled high-voltage power supply for CHAWS. During this first year Amptek designed, constructed, tested, and delivered a high voltage DC to DC converter that is used to sweep the CHAWS cylindrical housing through potentials from zero to minus 5000 volts. This supply can source up to 10 mA at 5000 volts; an output of 50 watts.

The completed flight unit of this high-voltage supply was delivered during the third quarter of 1992. It has been returned to Amptek several times to incorporate changes to accommodate the CHAWS system. The flight unit underwent several revisions including adding a suppression diode to an input coil, adding a temperature sensor, and finally adding a filter to the low-voltage supply portion to prevent EMI problems that occurred when the supply was fully loaded at maximum output. Each repair and/or modification required the retesting and requalification of the high-voltage supply board. The final version of the high-voltage circuit has been delivered to Phillips Laboratory and is being integrated along with the rest of the CHAWS hardware. Figure 21 is the current and latest schematic of the high-voltage circuit.

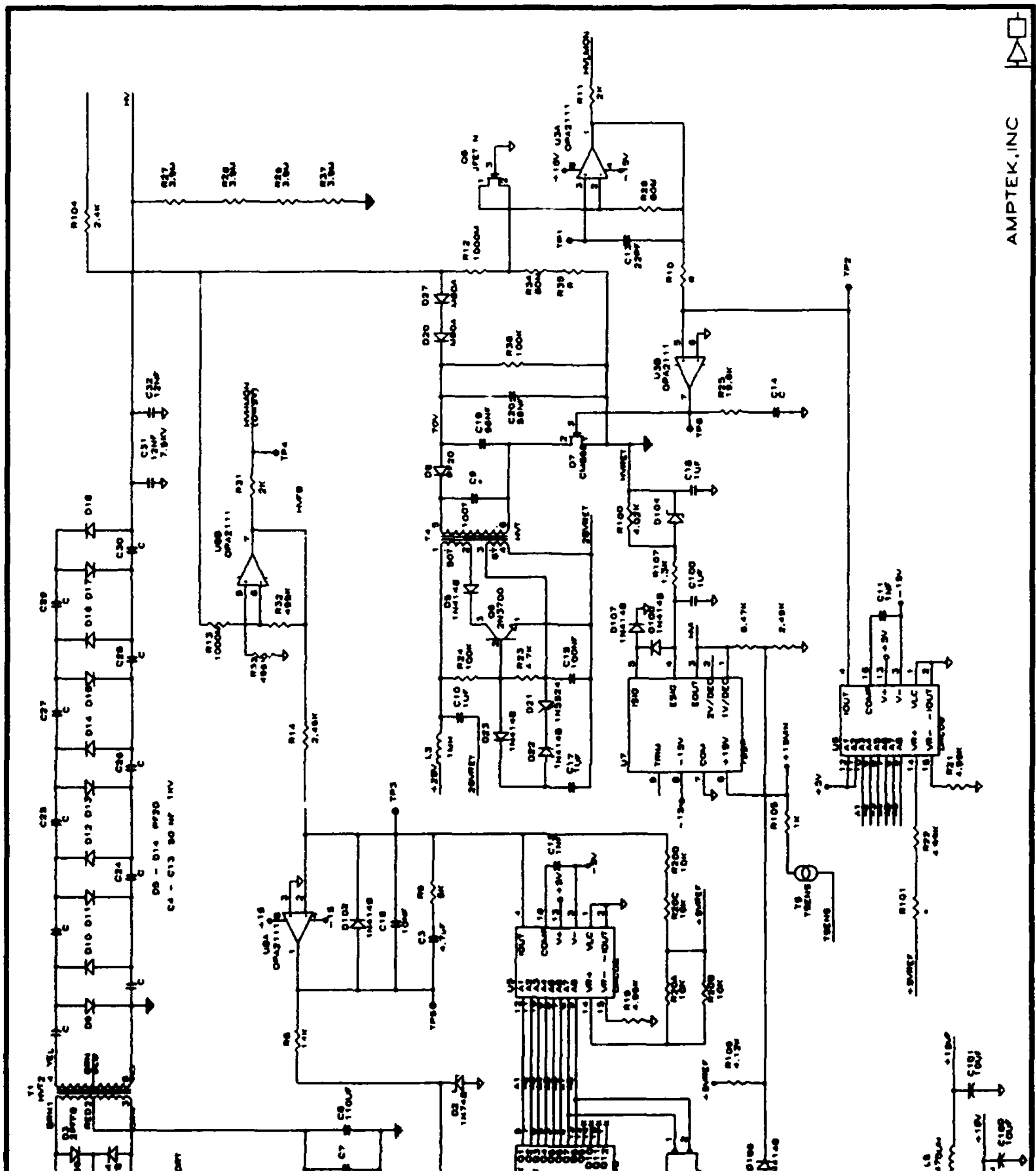


Figure 21. CHAWS Flight High Voltage Circuit

6.0 TASK #3

Task #3 efforts to date has been limited to rendering assistance to Phillips Laboratory in its preparation for upcoming participation in an international collaborative experiment, Waves in Space Plasma (WISP). Phillips Laboratory's portion of this experiment is called Space Wave Interactions with Plasmas Experiment (SWIPE) and consists of a suite of plasma diagnostic instruments, including electrostatic energetic-particle analyzers and a particle correlator system designed to detect and quantitize wave/particle interactions that result in density modulations of the charged-particle populations. The electrostatic analyzer and particle correlator system is to be patterned after the Shuttle Potential and Return Electron (SPREE) experiment built by Amptek for the Tethered Satellite System flown on the Space Shuttle STS-46 flight.

In this first year, the definition of the instrument suite that SWIPE will contain and the development of joint flight objectives with the Canadians (to utilize these diagnostics) as pursued and is now virtually complete. As presently baselined, the SWIPE system will consist of two pairs of nested triquadraspherical electrostatic analyzers (ESA s) to measure ions and electrons between 10 eV and 10 keV over a 180° field of view, a particle correlator system to process the ESA outputs, a pair of Langmuir probes to measure thermal plasma characteristics, a DC electric-field measurement system, and a multi-axis AC Magnetometer. All of these instruments will be controlled and scheduled through a Data Processing Unit (DPU), which Amptek will design and build.

Amptek provided both support and technical direction to Phillips Laboratory as the WISP experiment has evolved. We have attended many planning and development meetings, both at PL/GPSP at Hanscom AFB and Goddard Spaceflight Center in Maryland. We have also engaged the services of a British particle-correlator expert, Dr. Paul Gough, of the University of Sussex in Great Britain. Dr. Gough has been a co-investigator on the SPREE experiment and has begun to develop the necessary changes needed to modify the SPREE particle-correlator design into an instrument suitable for the WISP mission.

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